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SUMMARY

Solar electric propulsion (SEP) is an alternative to chemical and nuclear powered propulsion systems for both piloted and unpiloted Mars transport vehicles. Photovoltaic solar cell and array technologies were evaluated as components of SEP power systems. Of the systems considered, the SEP power system composed of multijunction solar cells in an ENTECH domed fresnel concentrator array had the least array mass and area. Trip times to Mars optimized for minimum propellant mass were calculated. Additionally, a preliminary vehicle concept was designed.

INTRODUCTION

NASA is currently assessing options for Lunar and Mars exploration missions. As a part of this assessment, various propulsion schemes are being evaluated. Solar electric propulsion (SEP), along with nuclear electric propulsion (NEP), nuclear thermal rockets (NTR), and chemical systems using aerobrakes (chem/AB), are the most likely candidates for use on a Mars transport vehicle (ref. 1). In this report, we evaluate SEP power systems and present a conceptual design for a Mars SEP transport vehicle.

There are a number of advantages anticipated for SEP systems. In summary, they are as follows: (1) SEP systems would probably have low development costs compared with other systems; (2) SEP systems are non-nuclear and, therefore, would not be a radiation hazard to equipment or personnel; (3) electric propulsion systems require less propellant than chemical propulsion systems, causing a high payload mass fraction of approximately 50 to 60 percent; (4) SEP power systems would be highly reliable; (5) SEP systems could be developed early in the next decade; (6) SEP power systems could be used in innovative ways after mission completion; and (7) SEP systems at higher power levels would have trip times to Mars competitive with chem/AB systems. Several of these advantages are discussed further in the following sections.

Cost

The cost of developing a SEP system is anticipated to be comparable to or less than other Mars transport propulsion systems for the following reasons:

(1) Photovoltaic (PV) power systems are flight-proven hardware with a well established performance database for small arrays.

(2) Space Station Freedom (SSF) will use large arrays (although an order of magnitude smaller in area than a SEP power system, still the largest arrays that will have been flown by that date) generating additional database information and much required practical experience.

(3) The solar electric propulsion stage (SEPS) concept was taken through the design stage and an engineering model was built (corresponding to the NASA Office of Aeronautics, Exploration and Technology (OAET) technology readiness level 6).

Reliability

The power and propulsion systems of a SEP vehicle would be designed to be highly reliable. The array wing module circuits could be designed so that damage to any localized section would not result in catastrophic power loss. The propulsion system would include multiple ion thrusters on multiple array masts. In the unlikely event that the thrusters on a mast are disabled, sufficient thrust could be generated by the remaining thrusters to return to Earth or continue on to Mars.

Specifically, there are four strategies that could enhance mission completion in the event of electric thruster failure: redundant thrusters, thruster overdesign, power shunt, and voltage increase. In the redundant thruster strategy, extra thrusters would be carried along as replacements. As a thruster is disabled, a spare thruster would be immediately substituted with no change in trajectory or trip time. In the second strategy, instead of adding thrusters, the thruster grid area would be increased. This permits increased power input into the thrusters at constant specific impulse (Isp). The thrusters in this case would be nominally run below maximum thrust performance. This increases thruster lifetime and does not alter Isp or thruster efficiency. Should a thruster fail, the performance of the remaining thrusters could be increased, keeping trajectory and trip time nominal. Both of these strategies have the disadvantage of carrying extra thruster mass, thus lengthening the nominal trip time compared with the case of no thruster redundancy. In the third option, the power into and the Isp from each thruster would be kept constant in the event of thruster loss. The excess array power could be rejected to space or the arrays could be feathered to reduce input power. With fewer thrusters operating and the others at the nominal specific power, the trajectory would be changed and the trip time lengthened. In the fourth strategy, voltage to the remaining thrusters would be increased. This increases Isp and thruster efficiency and decreases net thrust. This strategy requires a new trajectory and trip time would increase. The nominal trip time for the latter two strategies would be shorter than for the first two with no propulsion component failures. Changing the vehicle trajectory, however, may make rendezvous with Mars or an abort return to Earth problematic. It is beyond the scope of this study to perform failure analyses

for the power and propulsion systems. It is assumed that redundant thrusters would be used.

Reusability

The power system for a SEP vehicle could be used at Mars or the Moon for purposes other than propulsion. In a one-way Mars cargo mission in which the vehicle would be considered expendable, the PV arrays could still provide power after the vehicle achieves orbit. For a power system providing 5 MWe to the thrusters at 1 A.U. (earth orbit beyond the radiation belts), the arrays could provide at least 2 MWe at Mars. Such large power levels available in orbit could make power beaming to the surface a possibility. Power from the arrays could be transported to Phobos or Deimos via a conducting tether connected to the vehicle in a matching orbit a few hundred meters away. It may be possible that the PV blankets could be repackaged and transferred to the surface.

SEP systems could also be reused by flying them on multiple round trip missions. The vehicle could remain outside the Van Allen radiation belts on its return in order to avoid additional radiation damage. This suggests two future trade studies. First, would it be more efficacious to refurbish and resupply returning SEP vehicles at GEO and thus avoid radiation damage, or return the vehicle to a LEO resupply node thus reducing substantially the Earth-to-orbit (ETO) mass. Second, at what point would it be better to launch, assemble, and supply a completely new vehicle rather than accept the degraded performance of a SEP vehicle brought repeatedly through the Van Allen belts.

This study examines SEP vehicle design with emphasis on a key element - the power generation system. This study will also form the basis for future work in the Space Exploration Initiative (SEI) study process. Individuals who contributed to this study represent the following NASA Lewis organizations: The Advanced Space Analysis Office, the Power Technology Division, and the Structures Division.

METHODOLOGY

It was the purpose of this study to determine a set of potentially viable power systems for a SEP vehicle and to determine a possible vehicle configuration. The first step established a set of ground rules and figures-of-merit (FOM) by which potential power systems could be evaluated. Next, potential cell and array technologies were identified for the power systems. The technologies were evaluated and those that violated the FOM criteria were eliminated. Finally, selected technologies were combined into power generation systems and compared. From these systems, a possible vehicle configuration was generated. Throughout the latter part of this process, trip times optimized for minimum propellant mass were calculated as part of the iterative process of sizing the vehicle and its arrays.

Power System Ground Rules

The SEP vehicle was assumed to be an unpiloted cargo vehicle on a one-way trajectory to Mars, similar to a power system study for NEP vehicles (ref. 2).

There it would deliver a 400 metric tonne (MT) payload to a Mars synchronous orbit. The power system would be sized to provide 5 MWe to argon ion thrusters at 1 A.U. (after radiation degradation). Energy storage systems were not included in the power system because they are massive and unnecessary once the vehicle is beyond earth's shadow. It was assumed that the vehicle would coast when in shadow. The vehicle would be constructed in LEO; this assumption greatly affected the selection of technologies. In addition, two round-trip piloted Mars SEP vehicles were studied briefly - a 10- and a 28-MWe vehicle based on the 5-MWe cargo vehicle. These two vehicles are described in a later section.

The figures-of-merit used to select viable power generation technologies for the SEP vehicle were determined largely by assuming a LEO departure point. This is the result of two physical factors: the atmospheric drag at LEO and the trapped particle radiation in the Van Allen belts. The atmospheric drag requires minimal array area, requiring the selection of high efficiency cells. The particle radiation through which the vehicle must pass degrades the power output of the solar cells. Cells to be considered should therefore be highly radiation resistant. Since no solar cell is completely radiation resistant, the array must be oversized to provide the required 5 MWe after degradation. Oversizing the arrays, however, increases atmospheric drag. When the vehicle enters Earth's shadow, the atmospheric drag in LEO will cause the vehicle to lose orbital altitude. Energy storage could supply the electric thrusters with enough power to maintain orbit against the drag force. Excess power from the oversized arrays could be used to recharge fuel cells or batteries. However, the energy storage device would add tens of metric tonnes to the vehicle mass. Furthermore, the inherent inefficiencies of the storage system would waste energy that could be better used with redundant or oversized thrusters. Using, rather than storing, energy when it becomes available will more quickly raise the vehicle's orbit and simultaneously reduce drag and shadow time. Atmospheric drag or shadow region effects are not expected to seriously affect the vehicle. At worst, the spiral out trip time will be lengthened.

Atmospheric drag can be reduced by feathering the array wings. However, sun-pointing requirements may limit the degree to which planar arrays may be feathered; concentrator arrays cannot be feathered. Moreover, the continual readjustments that must be made to planar arrays as the vehicle spirals out from Earth would impose additional requirements on the structure, propulsion, and power systems. It was assumed in this study that the arrays would be continuously sun-pointing. It was found that the thrust available (during the sunlight periods) would more than make up the drag force and loss of altitude in the shadow region.

Another important figure-of-merit is power system mass. Propellent mass increases with increased power system mass (including structure). Increased mass could have several effects: it could increase trip time, it could displace payload mass, or it could do both. Therefore, it is important to find structural and power systems which are lightweight. The power processing unit (PPU) adds to the power system mass. This mass could be significant, since it makes up 25 percent or more of the total power system mass. It was assumed that the PPU specific mass would be 3.35 kg/kWe, which includes the power processing unit and radiators, and that it would have an efficiency of 90 percent.

Propulsion System Ground Rules

In addition to determining ground rules for the SEP power system, some assumptions were required for the thrusters. Argon ion and hydrogen magnetoplasmadynamic (MPD) thrusters are the most likely candidates for high power applications. For the initial vehicle study, ion thrusters were used because their performance is better understood and they are thus easier to model. It was assumed that the vehicle would utilize eight 1- by 5-m thrusters for the reasons given below.

Current experimental thrusters are small (10 to 50 cm in diameter) and require modest amounts of power (1 to 30 kWe). To keep system complexity to a manageable level, however, thrusters on SEP vehicles would have to be many times larger and able to handle megawatt power levels. These advanced thrusters would require technology advances in grid manufacture and materials that would support the large grid area required at high power levels. Thruster efficiency at high power levels and with large grids was assumed to be comparable to the efficiencies obtained with current experimental thrusters on the basis of the physical principles governing ion thruster performance. As with smaller thrusters, dissipated waste heat from the thruster would be radiated into space by the thruster grids' surface area.

To achieve the level of thrust required, it was calculated that 40 m^2 of grid area would be needed. By dividing the grid area into 8 1- by 5-m thruster sections, two benefits would result: (1) the probability of mission failure due to single thruster loss would be reduced compared with using fewer thrusters and (2) the power and propellant management system would not be overburdened by using too many thrusters. Because each thruster has an approximate lifetime of 10 000 hr, two sets of 8 thrusters each would be needed for a one-way trip to Mars; three sets (24 thrusters total) would provide 50 percent thruster redundancy. Additional thruster parameters are listed in table I.

Cell Technologies

Various cell technologies were evaluated for use on the SEP vehicle. Assuming a flight date circa 2014, many current research cells could reach maturation and be available for large-scale manufacture. Those examined were single-crystal silicon, amorphous silicon, gallium arsenide, indium phosphide, and III-V multijunction cells. A major evaluation criterion was the radiation damage that would be sustained by the cells in the Van Allen belts as described below.

Single-crystal silicon cells have been used almost exclusively in the United States' space program for the last 30 years. Space Station Freedom will use state-of-the-art 8- x 8-cm silicon cells. At the expected operating temperatures of the SEP array, silicon cells should have efficiencies of up to 13.5 percent. Since other cell technologies have higher efficiencies and better radiation resistance, silicon cells were not considered further.

Amorphous silicon cells, because they are less than 1 mil thick, are very lightweight. They may also be as much as 97 percent radiation resistant (i.e., the power output will be reduced by 3 percent due to radiation damage), as indicated by preliminary data. However, because amorphous silicon cells

have a relatively low conversion efficiency (~10 percent), SEP vehicles using these cells would necessarily have large array areas. For missions in which area is not a factor, amorphous silicon would be a good cell technology candidate. In this study, however, array area is a factor because of atmospheric drag; therefore, amorphous silicon cells were not considered further.

Gallium arsenide (GaAs) is a cell made up of elements from the third and fifth periods of the periodic chart - commonly known as a III-V cell - with a bandgap near the optimum for sunlight conversion. This yields a one-sun efficiency near 20 percent. For the SEP array, a conservative assumption was made of 18 percent efficiency at the operating temperature for production line cells. This is about the highest efficiency for single-junction cells. GaAs is also slightly more radiation resistant than silicon. A major disadvantage to a GaAs SEP array is its mass. GaAs is more than twice as massive as silicon, and it cannot be thinned to 2.5 mils as can silicon because it is too brittle. Current GaAs cells can be no thinner than 8 mils. A very promising solution to the mass problem is to grow the active GaAs cell layers, which are less 1 mil, on 3 mil germanium (Ge) substrates, cutting the cell mass by half. The technology to manufacture large quantities of GaAs on Ge production line cells (i.e., 5000 cells per month) is being developed under an Air Force manufacturing technology program. For the SEP mission, GaAs on a germanium substrate (GaAs/Ge) is a viable candidate for both planar and concentrator arrays. However, late in the analysis it was found that GaAs/Ge, albeit a viable choice, was not mass competitive with other cell technologies. Therefore, a GaAs/Ge SEP array is not among the final selected cell technologies. This technology should be considered in future SEP studies wherein the SEP vehicle does not traverse the Van Allen belts. The same performance parameters were assumed for the GaAs/Ge as for the 8 mil GaAs.

Indium phosphide (InP) is also a III-V cell with a high efficiency (17 percent for production line cells) and high radiation resistance (possibly 97 to 99 percent). NASA is currently experimenting with InP to determine the degradation of power output due to radiation damage. These experiments include both electron and proton damage at a variety of energy levels, the potential for illuminated annealing, and the applicability of using InP cells in concentrator arrays. Because current substrates for InP are expensive, the InP cells are assumed to be grown on a silicon substrate, currently the subject of a NASA contract (ref. 3). Because of its high efficiency and radiation resistance, InP was selected as a candidate for the SEP power system.

Multijunction cells use the solar spectrum more effectively than single-junction cells by having different spectral bands absorbed by different bandgap cells. The higher energy bandgap cell is placed above the lower energy bandgap cell. The short wavelength light is therefore absorbed in the upper cell with the longer wavelength light passing through to the lower cell. The two junctions may be series connected with required matching currents, or mechanically stacked with separate wiring harnesses. The major advantage is much higher efficiency, with performance levels of between 25 and 30 percent possible. Due to such high efficiency, the multijunction cell was also selected as a candidate SEP cell technology.

The various cell technologies listed have differing levels of radiation resistance. To calculate the radiation degradation of the cells as they pass

through the Van Allen belts[1], the concept of 1-MeV electron equivalence was used. The actual damage to photovoltaic cells by electrons of any energy can be related to the equivalent damage produced by 1-MeV electrons. In a similar manner, all proton damage is related to the equivalent damage caused by 10-MeV protons. A single 10-MeV proton causes the equivalent damage of 3000 1-MeV electrons (ref. 4). This allows degradation to be calculated by using the readily available 1-MeV experimental data on solar cell damage.

The JPL Solar Cell Radiation Handbook has several tables of 1-MeV annual equivalent fluences (ref. 5). Each table is for a particular orbital inclination and incident particle (electrons or protons). In each table, the annual equivalent fluence is given at 34 different altitudes (LEO to GEO) and eight different quartz coverglass thicknesses (0 to 60 mils). These tables summarize the results of the integration of electron and proton energies over different orbits along with experimental data of radiation damage. For III-V cells, one major change is the equivalence factor between 1-MeV electrons and 10-MeV protons. A factor of 1000 is considered a much better correlation and is used in this study.

Figure 1 shows the annual 1-MeV electron equivalent fluence as a function of altitude for three different coverglass thicknesses. This particular set of curves is the equivalent fluence for proton-caused damage. There is a significant reduction in fluence due to the coverglass. The reduction would be different at various altitudes due to the changing proton energy spectrum as a function of altitude.

Because of the amount of time spent in the proton-dominated radiation belts, more than 96 percent of the equivalent fluence is due to the protons. For the portion of the mission beyond GEO, the solar flares are the only significant contribution. Hence, to calculate the total equivalent fluence for the LEO to GEO portion of the SEP vehicle orbit, the fluence levels were summed for each day at a particular altitude over the range of altitudes. This was done for several coverglass thicknesses. The SEP array will not be a body mounted array with "infinite" back shielding; therefore, contributions from both front and back irradiation were included. It was assumed (for purposes of calculating radiation damage) that the orbit was at a 30 degree inclination, approximating the orbital inclination of Space Station Freedom.

The results of the above calculations give equivalent 1-MeV electron fluences for the LEO to CEO portion of the SEP vehicle orbit for any combination of front and back shielding thicknesses. With the data for 1-MeV solar cell degradation, one can plot the loss in power as a function of total thickness (fig. 2). These data can then be used to determine array sizes using various cell types and shielding thicknesses. It was assumed that the multijunction cells have the same degradation data as the GaAs since they are both III-V cells. This is the best assumption that can be made at present as there are very little data on multijunction cells.

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[1]Although solar flares are a source of radiation damage, for the SEP vehicle the predominant source of solar cell damage is the Van Allen radiation belts, consisting of both electrons and protons, each with a spectrum of energy levels.

Array Technologies

In this study, we considered both planar and concentrator arrays for the SEP vehicle. Planar arrays, used for over 30 years, are now using thinner cells on lightweight substrates such as Kapton. NASA is currently funding an effort to optimize lightweight arrays through the Advanced Photovoltaic Solar Array (APSA) program (ref. 6). One goal of the program includes achieving an array-specific power of greater than 130 W/kg (BOL) for array sizes in the 10- to 25-kW range. Since this design is the most advanced of any planar array design, it was decided upon as the baseline for the SEP array with the following assumptions: (1) the specific power of the APSA array would be constant to multimegawatt levels, (2) different cells could replace the current baseline 2.5 mil silicon cells with the proper changes in cell efficiency and weight, and (3) extra shielding beyond the 2 mils provided by APSA on both front and back sides could be added with the proper weight additions. (However, there is probably a limit as to how much shielding could be added, after which the array structure would need to be redesigned to support a greater load.) Using these assumptions, SEP power systems could be configured for APSA arrays with any cell type and with additional shielding for the trip through the radiation belts.

Concentrator arrays have no flight experience. Several concentrator arrays are being designed for military purposes, such as the TRW cassegrainian array, but they would be too massive due to their survivability requirements. Alternatively, NASA Lewis has a contract with ENTECH to develop designs for lightweight, high efficiency concentrator systems (ref. 7). These systems use domed fresnel lenses with high optical throughput (transmissivity greater than 90 percent) and lightweight structures. Currently, materials are being selected that will enhance the array's stability in a space environment.

RESULTS

From the cell and array technologies considered, four potentially viable SEP power systems were identified. They were compared to each other and a single system was proposed. Vehicle performance for all four systems was calculated and a vehicle configuration was developed.

Power Systems

The two array and two cell technologies selected were combined into four power generation systems:

(1) InP cells on an APSA array (InP-APSA).

(2) InP cells on an ENTECH concentrator array (InP-Conc).

(3) Multijunction cells on an APSA array (MJ-APSA).

(4) Multijunction cells on an ENTECH concentrator array (MJ-Conc).

Radiation damage factors were calculated for each of these systems. This factor was then used to scale the power systems to the mass and area required to achieve 5 MWe after radiation degradation (fig. 3). A breakdown of the power system mass by component is represented in figure 4. Note that the

power processing unit (PPU) mass ranges from 28 to 41 percent of the total power system mass. The PPU efficiency was calculated to be 90 percent with a specific mass of 3.35 kg/kWe.

The radiation belts degrade the power output of solar cells. Therefore, the SEP arrays must be sized to provide sufficient power initially such that after degradation the required power level is available. If this initial excess power were to be supplied to the thrusters - along with additional propellant - a higher thrust could be achieved thereby accelerating the vehicle more quickly through the radiation belts. By doing this, less radiation damage would be sustained by the cells and less array area would be needed. There is, therefore, an optimum power level which will be different for each type of power system. It was beyond the scope of this study to determine the optimum power levels for each system; however, several iterations were performed. Figure 5 shows the initial power that the arrays must supply at an orbit of 408 km (Space Station Freedom's orbital altitude), assuming an initial acceleration of 6.45 MWe, such that 5 MWe would be supplied once through that region.

Vehicle Performance

The vehicle trajectory was analyzed by a program which optimizes thruster steering and launch/arrival dates to minimize the propellant required to reach Mars. The specific mass of the power system was varied parametrically to obtain the results shown in figure 6. This graph shows the effect that the power system specific mass has on the initial vehicle mass in LEO for various trip times. Analytic spiral approximations were used for the escape and capture portions of the trip. Coasting periods of lengthening duration were included in the analysis, increasing trip times while lowering vehicle initial mass.

The vertical lines in figure 6 represent the specific mass of the four power systems examined. This figure indicates that a SEP vehicle using multi-junction concentrator arrays could be designed to reach Mars in 1000 days with a total vehicle initial mass in low earth orbit (IMLEO) of 644 MT (60 percent payload fraction). By following an all-propulsive trajectory, this same vehicle could reach Mars in 868 days with a mass penalty of 34 MT.

Vehicle Design

Figure 7 depicts a possible configuration for a SEP vehicle. The octagonal array area would be composed of eight keystone-shaped split blanket arrays on eight radial masts. Thrusters would be placed on the extended ends of each of the masts. The argon propellant tanks and 400 MT payload would be positioned in the center.

The octagonal shape provides a four-fold symmetry and a logical structural framework for the split blanket arrays. It also provides modularity and a relatively short distance for the power management and distribution (PMAD). Two benefits are gained by positioning the thrusters at the end of the mast extensions: (1) moment arms are lengthened, thereby enhancing steering capability and (2) the thrusters are spaced apart, thus decreasing the probability of a disabling propulsion system loss. If all the thrusters were clustered together, a single micrometeoroid or particle impact could disable a

significant number of thrusters and cause mission failure. The four-fold symmetry allows for a greater flexibility in steering by permitting selective firing of thrusters arranged on the eight radial masts. A severed propellant line from a meteoroid impact could take at most one-eighth of the thrusters out of service in this configuration. Mission survivability would be improved by designing multiple array wings and mast sections.

The SEP vehicle mast design is the two-wing center mast design used for both the Space Station Freedom and APSA designs. The main differences between these and the SEP vehicle is that the SEP vehicle design is much larger and the top cross member is longer than the bottom cross member (nonrectangular blankets). It does not appear that either of these differences would present complications, although the design of the cross members and boom probably will differ from the previous designs (e.g., continuous beams for the APSA versus post tensioned truss structures for the SEP vehicle) because of the large dimensional dissimilarities. It presently is not clear exactly how the structural components would look, but it is expected that the mast and boom cross sectional areas would increase and the other components would remain relatively similar in comparison with the smaller spacecraft. The overall design would most likely be constrained by a lower stiffness limit (e.g., 0.01 Hz) and a demand to withstand a maximum acceleration level (e.g., 0.1 g). For a structure of this large a size, post tensioned guy wires could possibly be used for providing stiffness and a means for vibration suppression. Cables may also be used to minimize the blanket substrate thickness (and mass) by relieving the substrate from having to carry all of the blanket tension loading. In the future, a study could be made to investigate the tradeoff between mast length, weight, and thruster requirements.

DISCUSSION

Solar electric propulsion is a viable transportation technology for Mars missions. Because electric propulsion vehicles would have the capacity to transport large payloads, they would have a natural application as cargo vehicles. Although piloted SEP cargo vehicles are possible, unpiloted cargo missions may be more practical due to the long trip time to Mars and the large power requirement for a piloted mission. Nevertheless, two piloted SEP vehicles were analyzed which were sized to have competitive round trip times with chem/AB vehicles. The two vehicles are a result of identifying two trajectories starting at geosynchronous earth orbit (GEO) - outside the Van Allen belts - flying a high-energy trajectory to an areosynchronous orbit, and returning to GEO after a short stay time. The first trajectory would require 28 MWe and would take the vehicle to 0.72 AU perihelion (nearly the orbit of Venus) on its way to Mars. The other trajectory would only require 10 MWe from the power system, but it would take the vehicle to within 0.5 AU perihelion. For both vehicles, the outbound payload mass was 120 MT and the GEO return payload mass was 40 MT. On the basis of a 5-MWe cargo vehicle design, both vehicles had significantly increased array areas to increase their power output. By increasing power levels and thruster grid area, higher thrust levels can be produced thus lowering trip time. A comparison of the three different SEP vehicles is given in table II. Figures 8 and 9 show one wing of eight for the 10 MWe and the 28 MWe vehicles, respectively, along with the central cargo area and dimensions. The eight wings would be deployed radially as in figure 7.

RECOMMENDATIONS

Further study should be made of SEP vehicles and their power systems for Mars transportation applications. Because specific and somewhat arbitrary ground rules were made for this reference study, future studies could analyze power systems based on different assumptions, such as smaller payload masses or reduced or increased power levels. Future studies could also investigate the effect of energy storage for orbit maintenance on the power system mass and area. In addition, studies may assume a Lagrange point or GEO return node as a staging point for multiple missions. Future studies could also optimize the power system such that excess power available before radiation degradation could be used to provide additional speed, thereby decreasing the required power system mass and area. The power processing unit should also be analyzed; by increasing PPU efficiency, the SEP power system mass can be reduced significantly. Also, trajectories could be identified that would optimize trip time or propellant mass. The propulsion system could be optimized, as well. Finally, future studies should include cost estimates.

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TABLE I. - THRUST PARAMETERS FOR SEP CARGO VEHICLE
5-MWe PROPULSION SYSTEM

Thruster type	ion
Propellant	argon
Thruster size, m	1 x 5
Thruster efficiency	0.67
Specific mass per thruster, kg/kWe	2.3
Number of thrusters running	8
Number of thrusters total	24
Power to thruster at 1 A.U., MWe	0.625
Voltage at thruster, Vdc	2000
Beam current, A	679
Ion production, W/A	150
Specific impulse, s	5000

TABLE II. - SEP VEHICLE COMPARISON

	Array area, m^2	Array mass, MT	Power system mass, MT	Alpha, kg/kWe	Initial power at LEO, MWe
5 MWe LEO departure					
Planar PV					
InP on 3 mil Si	39 512	47.5	77.4	15.49	6.57
MJ on 3 mil Ge	34 442	48.8	87.2	17.44	9.09
ENTECH conc.					
InP	24 980	31.1	58.4	11.68	5.92
MJ	22 674	28.3	61	12.19	7.53
10 MWe GEO departure					
Planar PV					
InP on 3 mil Si	62 346	59	68	6.76	10.36
MJ on 3 mil Ge	40 932	48	57	5.67	10.81
ENTECH conc.					
InP	44 160	59	67	6.74	10.47
MJ	32 548	43	50	5.05	10.81
28 MWe GEO departure					
Planar Pv					
InP on 3 mil Si	174 570	166	175	6.25	29.01
MJ on 3 mil Ge	114 608	135	144	5.16	30.26
ENTECH conc.					
InP	123 647	166	175	6.26	29.32
MJ	91 134	119	128	4.56	30.26

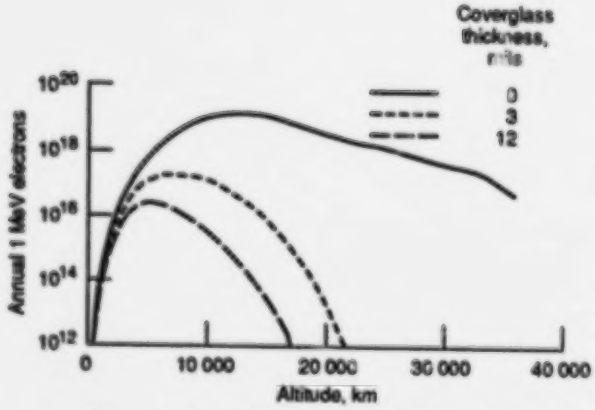

Figure 1.—Proton fluence (1 MeV electron equivalent) at 30 degrees inclination.

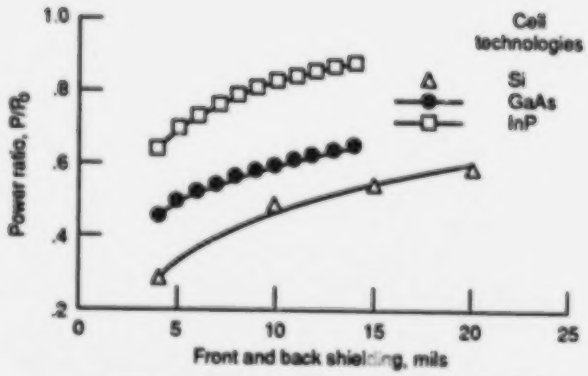

Figure 2.—Power ratio versus total shielding for SEP vehicles spiraling from LEO to GEO.

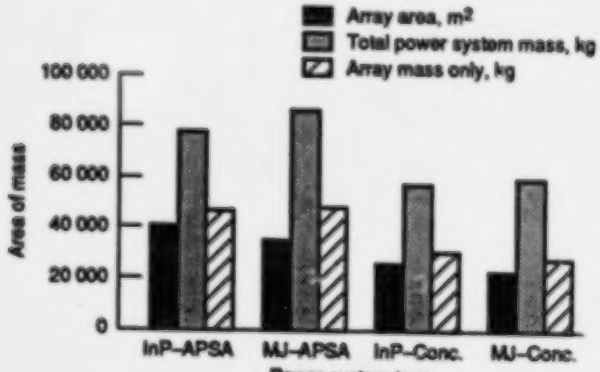

Figure 3.—Comparison of array and system mass and array area for the four 5-MWe SEP power systems.

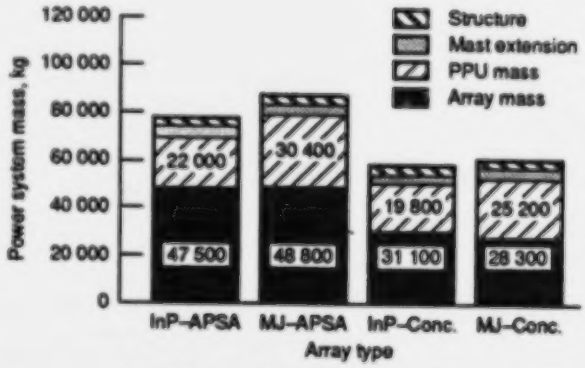

Figure 4.—Breakdown of component masses for the four 5-MWe SEP power systems.

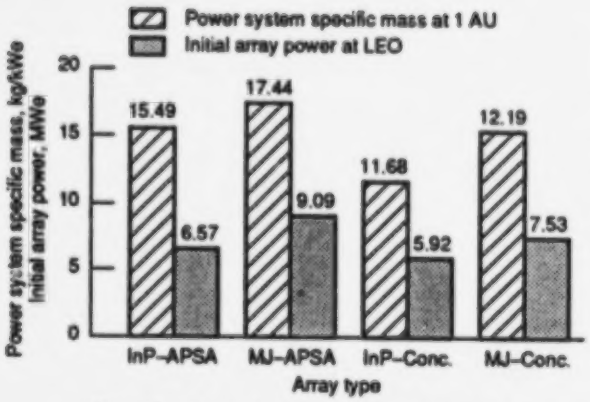

Figure 5.—Comparison of power system specific mass and initial array power for the 5-MWe class SEP cargo vehicle.

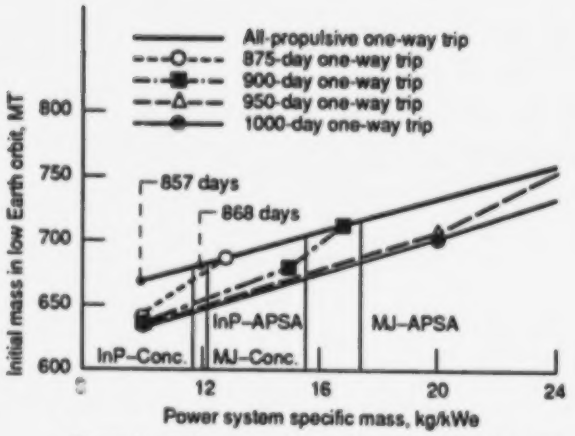

Figure 6.—SEP cargo vehicle performance (5-MWe class).

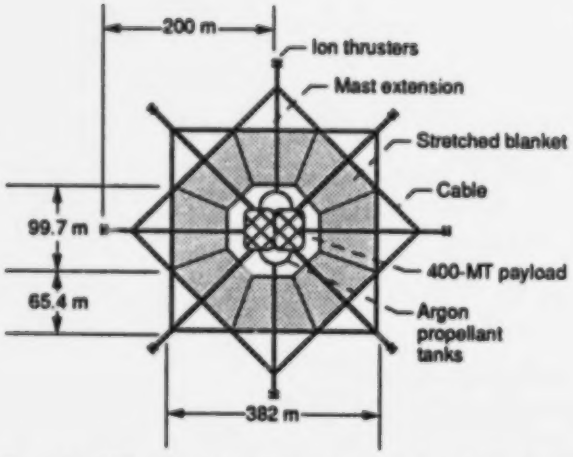

Figure 7.—SEP cargo vehicle configuration with a 38 000 m^2 array design (5-MWe class).

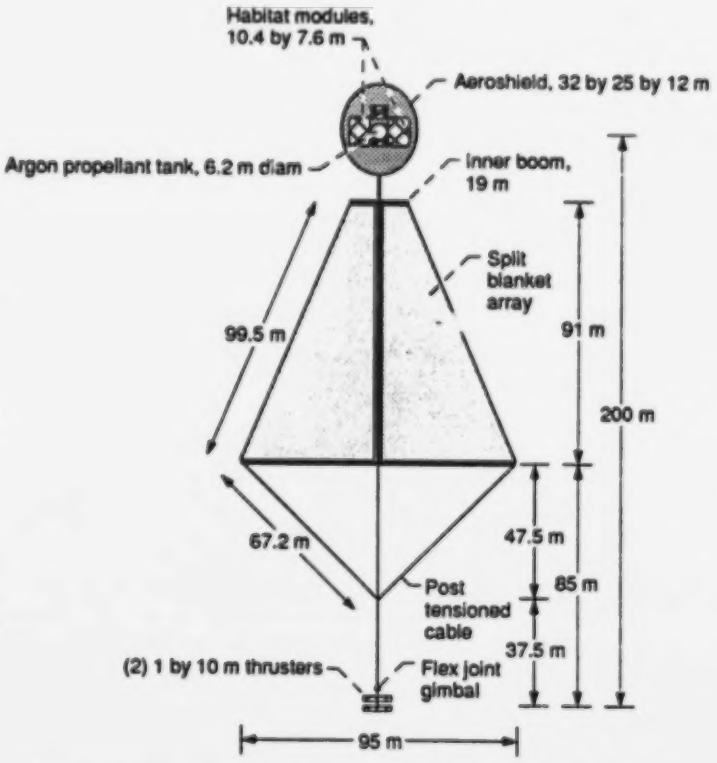

Figure 8.—10-MW SEP vehicle, single wing design. [One wing of 8; total array area, 42 000 m^2; array area per wing, 5 250 m^2.]

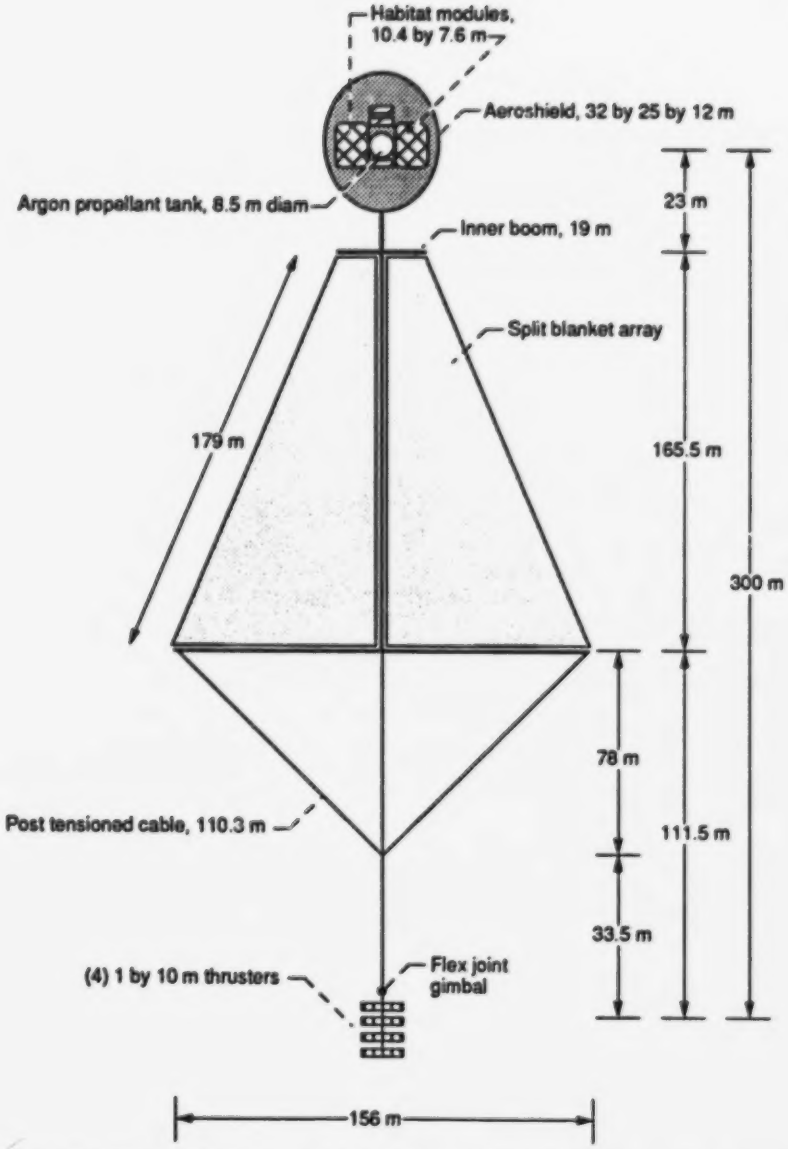

Figure 9.—28-MW SEP vehicle, single wing design. [One wing of 8 with cargo area shown; total array area, 116 000 m^2 (28.66 acres); array area per wing, 14 500 m^2.]

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16. Abstract

Solar electric propulsion (SEP) is an alternative to chemical and nuclear powered propulsion systems for both piloted and unpiloted Mars transport vehicles. Photovoltaic solar cell and array technologies were evaluated as components of SEP power systems. Of the systems considered, the SEP power system composed of multijunction solar cells in an ENTECH domed fresnel concentrator array had the least array mass and area. Trip times to Mars optimized for minimum propellant mass were calculated. Additionally, a preliminary vehicle concept was designed.

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